IAC–18–C1.9.7x45357

Double Asteroid Redirection Test (DART) Mission Design and Navigation for Low Energy Escape

Justin A. Atchison
The Johns Hopkins University Applied Physics Laboratory, United States, Justin.Atchison@jhuapl.edu

Matthew Abrahamson
The Jet Propulsion Laboratory, California Institute of Technology, United States, Matt.Abrahamson@jpl.nasa.gov

Martin T. Ozimek, Brian Kantsiper, Elena Adams, Andrew F. Cheng, Andrew Rivkin, Cheryl Reed
The Johns Hopkins University Applied Physics Laboratory, United States

Shyam Bhaskaran, Zahi Tarzi, Dianna Velez, Julie Bellerose, Frank Laipert, Daniel Grebow
The Jet Propulsion Laboratory, California Institute of Technology, United States

This paper describes the evolution of the NASA Double Asteroid Redirection Test (DART) mission design and navigation. Specifically, the mission has been conceived as (1) a hydrazine bus on a ballistic trajectory, (2) a low-thrust bus launching from a geostationary transfer orbit and spiraling to escape, and (3) a low-thrust bus that launches with a small positive escape energy. This paper discusses the rationale in favor of the third concept, low energy escape, and describes the key mission design and navigation studies. In an effort to be compatible with an unknown co-manifest partner, the trajectory design must account for a large range of launch energies, orientations, and dates. The navigation approach must account for sensitive regions in the trajectory and plan for both low-thrust and chemical phases of flight. These findings are relevant to other missions pursuing low-cost interplanetary rideshare concepts.

I. Introduction

The NASA Double Asteroid Redirection Test (DART) program in the Planetary Defense Office is designed to be the first mission to demonstrate asteroid kinetic deflection.[1,2] The spacecraft will use autonomous terminal guidance to impact the smaller member of the 65803 Didymos binary asteroid system. The mission leverages a rare 2022 conjunction with Didymos, when it passes within 0.07 AU of Earth. The asteroid is readily accessible, and the impact experiment is observable from Earth. The impact's effect is measured as a change in the Didymos light-curve, which relates the change in the binary's orbit period. Since this measurement is achievable from Earth during the conjunction, a second in-situ satellite is not required to achieve the primary mission objectives. However, this constraints the impact date to the time of the conjunction: late September 2022 to early October 2022. This causes mission schedule and trajectory time-of-flight to be key mission drivers.

The principal goal of DART's mission design is to transfer the spacecraft from Earth to the Didymos system with a specific arrival geometry that maximizes the experiment's observability. Where possible, it is also desirable to fly by a near Earth asteroid (NEA) to practice many aspects of the impact critical operations. That is, the terminal guidance algorithms and activities will be tested, with the key difference that the spacecraft will be commanded to passively fly by the NEA rather than impact it. The required Didymos impact conditions impose constraints on the arrival solar phase angle, the asteroid-Earth angle (for pointing DART's high gain antenna), and the angle of DART's relative velocity and the Didymos binary orbit plane. These constraints are detailed in Sarli et al, 2017.[3]

The Navigation team is responsible for quantifying the expected orbit knowledge and control during the interplanetary transfer, the opportunistic NEA flyby, and approach to the Didymos system. The spacecraft is tracked with radiometric ranging and Doppler measurements via the Deep Space Network (DSN). The asteroid encounters are navigated using optical images from the on-board imager DRACO (Didymos Reconnaissance and Asteroid Camera for

OpNav).[4] The final phase of the impact occurs using autonomous terminal navigation,[5] which is not discussed here.

Since it's conception, DART's mission architecture has evolved significantly. The primary trade-space is the launch energy required by the launch vehicle. DART has been designed over a range of initial launch energies: as low as geostationary transfer orbit (C3 ≈ -16 km^2/s^2) to as high as a direct ballistic transfer (C3 ≈ +6 km^2/s^2). Low launch energy cases are enabled by the addition of an ion propulsion system (IPS), which represents a significant design driver. This paper describes the mission evolution to date and addresses the key trajectory and navigation trade-studies associated with one candidate concept: the low energy escape case.

II. Mission Evolution

DART was initially conceived as low-cost spacecraft equipped with only mono-propellant hydrazine. In this concept, it followed a ballistic transfer to Didymos, where the launch vehicle provided the initial transfer energy and DART was only required to execute targeting corrections along the way. This approach is the lowest cost for the spacecraft bus, though it requires a dedicated launch vehicle. For the majority of launch dates, it doesn't include an asteroid flyby. Further, the arrival geometry cannot be meaningfully adjusted in flight, it is a product of the ballistic transfer solely.

In 2016, the DART program conducted a formal trade-study and decided to add an ion propulsion system (IPS): the NASA Evolutionary Xenon Thruster - Commercial (NEXT-C).[6] With the addition of the NEXT-C thruster, we were able to explore a wide range of possible launch scenarios. The mission especially considered opportunities to share the cost of a launch vehicle by coordinating DART's launch as a co-manifest mission. In this case, the launch vehicle ascent would be optimized to deliver two spacecraft to orbit, with DART "riding along" with a primary satellite. DART would use NEXT-C to achieve the remainder of its transfer to Didymos, making up for the energy not provided via a direct transfer. This represents an opportunity to trade the cost associated with IPS's added capability, with the cost of a dedicated launch vehicle. With the IPS, there are a variety of possible launch conditions from which DART can complete its mission, though each condition has effects on the spacecraft and mission design.

Figure 1 depicts the set of options we studied, each of which is described below.

II.i Direct Ballistic Transfer Option

The initial DART baseline design uses only mono-propellant hydrazine, resulting in a simple spacecraft with a wet mass of roughly 350 kg. This is known as the chemical-propellant option.[7] The heliocentric transfer is ballistic, and is provided by a dedicated launch vehicle. DART only executes small targeting maneuvers to correct for errors. In some very specific launch cases, it is possible to achieve a NEA flyby prior to the impact. This requires a small amount of additional ΔV to adjust the trajectory after the flyby to re-target Didymos. There are two primary launch periods for the chemical-propellant option: June to

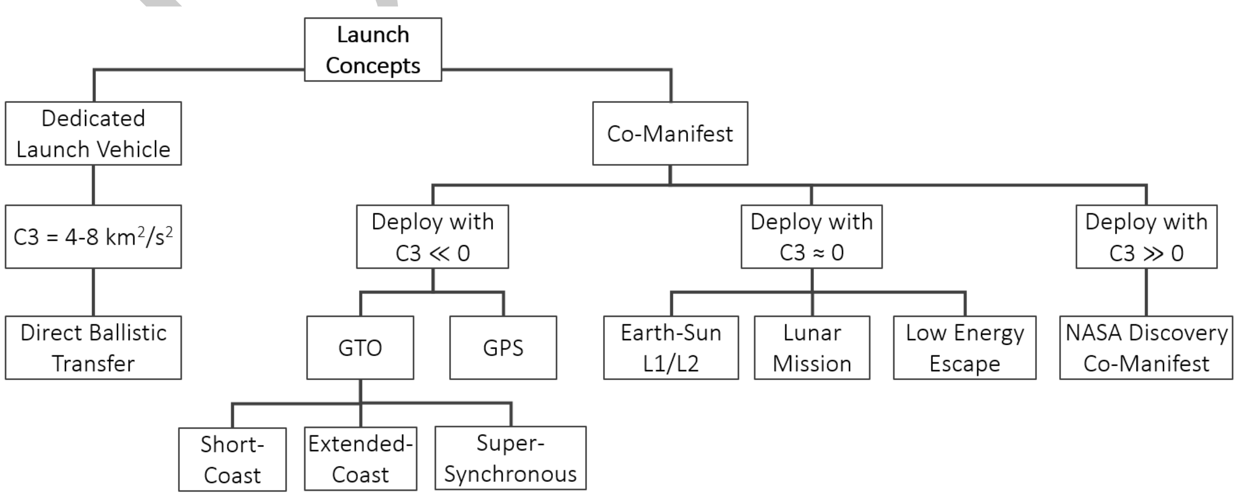

Fig. 1: DART Launch Tradespace

August of 2021 and November 2021 to February 2022. Just prior to the impact date, there is a final window from June to August of 2022, although launching in this period would represent an operational challenge with very limited time to prepare for impact. For all of the launches, the required declination of launch asymptote has a high magnitudes (∼60-70^o). This is because DART requires the launch vehicle to provide a non-trivial inclination change ($\tilde{3}.5^o$ with respect to the ecliptic) to reach Didymos at the time of conjunction. This option is the simplest, but requires a dedicated launch vehicle.

II.ii GTO Co-Manifest Option

With the addition of the IPS, the mission sought opportunities for co-manifest launches, where DART would launch into a standard captured Earth orbit and then use the IPS to spiral to escape. Geostationary Transfer Orbit (GTO) is a common launch case with appreciable launch energy. Broadly speaking, a GTO is any orbit which enables an energy-efficient transfer to geostationary orbit (GEO). Each GTO is optimized to the launch vehicle's and geostationary satellite's capabilities, so it can be difficult to generalize the orbit elements. The argument of perigee tends to be roughly 180^o. The right-ascension-of-ascending-node (RAAN) is associated with the launch time-of-day. This is mission specific, but is selected with considerations for operations and dynamical stability. For apogee altitude, perigee altitude, and inclination, we studied historical GTO launches from the US and defined the option-space using four classes:

- **Subsynchronous**: This is the lowest energy option, with a low perigee (typically <500 km) and a low apogee (roughly 25,000 km to 30,000 km). Because these are low energy and relatively uncommon, we eliminated them from our trade-space.

- **Short-Coast**: These GTO transfers have a low perigee (typically <1000 km) with an apogee near GEO. The inclination is roughly the launch site's latitude. They are called short-coast because the launch profile uses all the upper stage firings near perigee.

- **Extended-Coast**:These GTO transfers have a higher perigee (typically >2000 km) with an apogee near GEO. They are called extended-coast because the launch ascent includes a coast phase that is used to increase perigee altitude and reduce inclination.

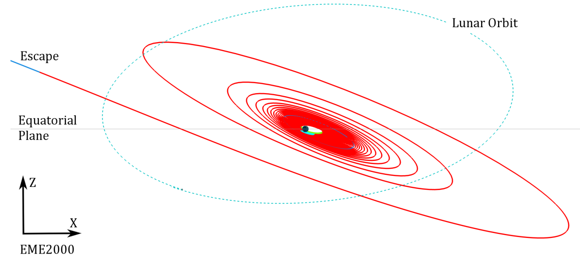

Fig. 2: Sample DART spiral escape from a short-coast GTO.

Case	Initial C3 (km^2/s^2)	Prop Used (kg)	Spiral Duration (day)	Latest Launch (D/M/Y)
Short-Coast	-16.3	88	249	05/04/21
Extended-Coast	-14.6	79	202	22/05/21
Supersynchronous	-10.9	70	178	15/06/21

Table 1: Summary of GTO Escape Cases

- **Supersynchronous**: This is the highest energy option. They are called supersynchronous because their apogees are above the (1:1) geosynchronous altitude. These cases are typically favorable for low-thrust transfers to GEO.

Figure 2 shows a sample spiral escape from a short-coast GTO. The initial GTO is the smallest elliptical orbit in green. The spacecraft thrusts continuously to raise orbit energy, with the exception of periods of eclipse or phasing orbits. This case represents a single case among many thousands of evaluations. For full mission flexibility, DART needs to be designed to launch at any time-of-day (any RAAN) over a meaningful launch period of many months. Each case requires an independent trajectory optimization.

Based on these simulations, we developed a DART design that operates NEXT-C at 3.48 kW with a wet mass of 680 kg.[8] Table 1 gives the associated maximum escape duration and propellant use over the different launch types, dates, and RAANs. The latest launch date is the first half of 2021, driven by the spiral-escape durations that are on the order of 6-8 months. Additionally, the spiral altitudes and duration drive the spacecraft design to be able to accommodate a relatively high radiation environment and many eclipses.

II.iii Low-Energy Escape Option

In this approach, the launch vehicle ascent is optimized to deliver the primary satellite into an operational orbit and then restart its upper stage to provide DART with a slightly positive escape energy. This option is enabled by partnering with satellite missions that have significant excess launch vehicle capacity or by augmenting existing missions with an additional launch vehicle solid rocket booster, modified upper stage, or non-coverable first-stage.

This approach is notionally compatible with any co-manifest orbit type. For example, DART could hypothetically be launched with a low Earth orbit (LEO) sun-synchronous partner. Following the deployment of the LEO satellite, the upper stage would restart and execute an escape burn. DART, the upper stage, and any associated debris would then be departing the Earth-Moon system passively. Once commissioned, the NEXT-C thruster would begin to shape the slow outbound drift, accelerating to reach the asteroid flyby and Didymos impact.

One limitation on co-manifest orbit type is the number of available upper stage restarts. This concept requires the upper stage to retain a restart for the DART escape, which may preclude some orbit types. For example, the launch profile for a direct geostationary orbit injection typically uses all available upper stage ignitions. Nonetheless, given that the low energy escape case is at least compatible with GTO launches, it would have as many or more potential partners than the GTO case.

Compared with the GTO approach, the main advantage of a low energy escape is that it eliminates the need for a long, many-month spiral escape. This offers schedule margin, simplifies the low thrust spacecraft bus by removing post-launch eclipses and reducing radiation exposure, reduces operations, and simplifies orbit debris compliance. Since the 6 - 8 month spiral is removed, the latest launch is extended to roughly 15 Oct 2021, including an asteroid flyby in March of 2022.

The challenge for this option is to envelope the range of possible launch conditions. To maximize co-manifest orbit compatibility, DART must be capable of completing its mission with an escape state that is directed nearly anywhere on a unit sphere. This analysis is discussed in Section 3.1.

II.iv Other Options

The remaining options in Figure 1 considered for a low-thrust spacecraft design are summarized below:

- **GPS**: We evaluated co-manifesting with spacecraft going to Global Positioning System (GPS) orbits. This case is favorable because it represents a relatively common high-altitude US launch opportunity. GPS satellites are delivered to circular 12-hour orbits at an altitude of roughly 20,200 km. However, from DART's perspective the driving parameter is orbit energy. The GPS orbit, though high altitude, only offers an initial C3 of -21.4 km^2/s^2, which requires significant additional thrusting over the GTO cases.
- **Earth-Sun Lagrange Points**: DART could co-manifest with missions to the Earth-Sun Lagrange points (L1 and L2). In these cases, DART is able to adjust the near-zero launch energy and achieve the Didymos transfer. The results are very similar to the Low Energy Escape case, with slightly more favorable performance because the direction of the transfer is favorable. Unfortunately, there are no candidate Lagrange point missions launching within DART's required schedule.
- **Lunar Missions**: It's possible for DART to co-manifest with a lunar mission. The implications depend heavily on the type of lunar transfer. Some missions transfer directly (e.g. Apollo Series, EM-1). Some missions use a highly elliptical phasing orbit (e.g. LRO, LADEE). Some missions use weak stability boundary techniques (e.g. Hiten, GRAIL). And some recent missions use on-board propellant to raise apogee from an initial GTO (e.g. SMART-1 or Chandrayaan-1). None of these cases is obviously incompatible with DART's design. DART would have the option of using hydrazine or xenon to shift its trajectory away from the host, enabling a lunar flyby to depart the Earth-Moon system. This would represent an additional critical event but would not drive the spacecraft design.
- **NASA Discovery Missions**: The NASA Lucy and Psyche Discovery missions are launching in the early 2020's. In each case, the launch energy is too high for DART to recover from, given the limited time to the Didymos encounter. So unfortunately, DART is incompatible with these options.

II.v Comparison of Options

Figure 3 gives a summary of the launch dates associated with the key options. The colors indicate where an opportunistic asteroid flyby is achievable. The data is directly correlated to the launch energy, with higher energy options offering later launch dates. Given DART's constrained impact date, later possible launches represent an advantage for the program.

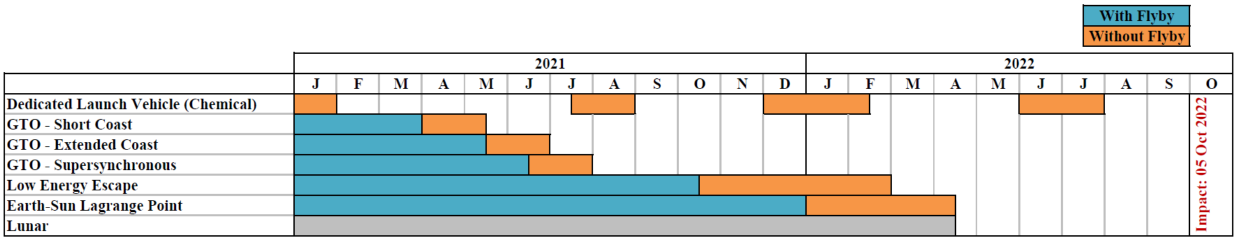

Fig. 3: Summary of Launch Periods for Different DART Launch Cases.

Launch Case	Latest Launch Date	Co-Manifest Compatibility	Flyby In Baseline	Adjustable Impact	Eclipses	Radiation
Direct Ballistic (Chemical)	Mid-2022	Not-Possible	Rarely	No	≤ 1	Low
GTO	Early-2021	High	Yes	Yes	Many	Very High
Low Energy Escape	Late-2021	High	Yes	Yes	≤ 1	Low
GPS	Late-2020	Med	Yes	Yes	Many	High
Earth-Sun Lagrange	Late-2021	None-Available	Yes	Yes	≤ 1	Low
Lunar Mission	Variable	None-Available	Unknown	Likely	≤ 1	Low
NASA Discovery Co-Manifest			Not-Possible			

Table 2: High Level Comparison of the Launch Options

Table 2 gives a high level comparison of the different options, listing the key advantages and disadvantages of each.

Based on this data, the DART program conducted a formal trade-study and decided to adjust from a GTO-compatible mission to a low energy escape mission. This simplified many spacecraft subsystems including power, which no longer needed to size a battery to survive consecutive long eclipses. The thermal design was simplified since the spacecraft's orientation with respect to the Sun is more stable without large variations associated with the spiral. The radiation shielding and required hardware testing was greatly reduced. And finally operations were simplified with slower maneuver design cycles.

The details for the mission design and navigation of this low-energy escape concept are given in the remainder of this paper.

III. Low Energy Escape - Mission Design

The baseline design for the low-energy launch concept is presented in Figures 4 and 5. Figure 4 shows the trajectory in a heliocentric inertial frame aligned with the mean ecliptic (EMO2000). The orbit of DART "leads" Earth and is inclined with respect to the ecliptic. The thrust periods and direction are depicted as red lines. The thrust is primarily oriented in the out-of-ecliptic direction to achieve the inclination change. There are three thrust arcs, totaling 137 days. There are up to 7 trajectory correction maneuvers (TCMs), with 3 prior to the flyby and 4 prior to the impact. The TCMs are nominally conducted using the monopropellant hydrazine propulsion system.

Figure 5 shows the same trajectory in an Earth-centered frame that rotates with the Earth-Sun system, such that the Sun is in the -X direction and the +Z direction is aligned with Earth's orbital angular momentum. In this frame, the trajectory's motion with respect to Earth is more clearly seen. DART launches in the direction of Earth's velocity (+Y) and Earth's momentum (+Z) and begins thrusting to move ahead of Earth and impart a plane change. The impact occurs in the ecliptic -Z direction ("below" Earth's orbit).

Many of the key trajectory values are given in Table 3. This trajectory launches on the first day of the launch period, 15 Jun 2021. DART operates NEXT-C at a fixed throttle level of TL28, with a duty cycle of 90% (5% allocated for operations and 5% unallocated as margin). DART launches with a mass of 636.5 kg, of which 50.8 kg of xenon is deterministically required for the interplanetary transfer. The flyby of 2001 CB_{21} occurs on March 06 2022, 6 months prior to the impact of Didymos on 05 Oct 2022. The impact occurs with an arrival solar phase angle (Sun-Didymos-DART) of 60^o, which is relevant for the image-based optical navigation. The impact

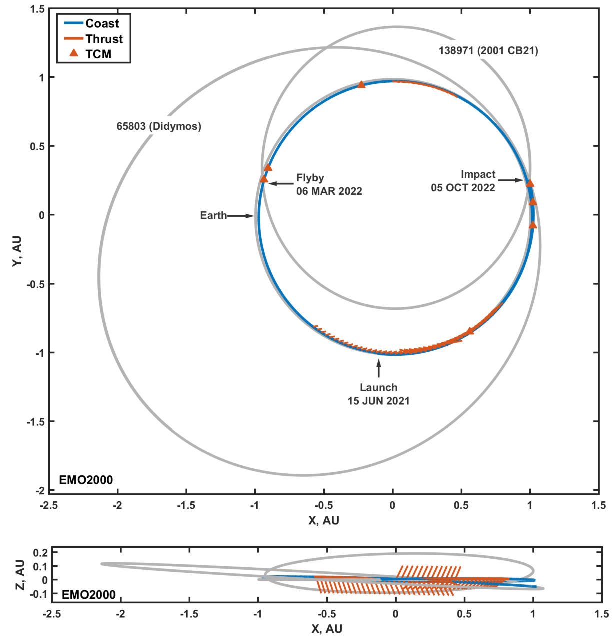

Fig. 4: Sample low-energy escape trajectory, given in inertial heliocentric coordinates.

speed is 6.0 km/s, 15.8^o out of the binary orbit's mutual plane, as shown in Figure 6.

Table 4 gives the mass budget to accommodate this trajectory as well as the full design space, discussed in Section 3.1. An allocation of 0.5 kg is associated with arrival timing adjustment. This is because the phase of the binary orbit is currently unknown, but will be observable post-launch, in early 2022.

III.i Key Studies

The baseline case presented above is a subset of the thousands of studied launch cases. The spacecraft must be designed to accommodate some relevant subset of the possible launches. Here, we present the key studies we used to define and quantify the available launch conditions and assign reasonable margins for IPS performance.

Launch Conditions

The low-energy escape is meant to be compatible with as many co-manifest partners as possible. From a trajectory standpoint, this means that the IPS should be capable of completing the full DART mission (flyby of 2001 CB_{21} and impact Didymos with the required arrival geometry) from any escape direction. That is, the escape direction associated with a polar LEO co-manifest partner would likely be different from the escape direction associated with a

Parameter	Value
Time of Flight	477.6 day
Deterministic Xenon	50.8 kg
Total Thrust Duration	137 day
NEXT-C	
Throttle Level	TL28
Thrust	137.1 mN
Specific Impulse	2943 s
Duty Cycle	0.90
Launch	
Date	15 Jun 2021
C3	0.1 km^2/s^2
Right Ascension*	330^o
Declination*	25^o
Mass	636.5 kg
Flyby	
Body	2001 CB_{21}
Date	06 Mar 2022
Velocity	11.8 km/s
Solar Phase Angle	23.5^o
Impact	
Date	05 Oct 2022
Speed	6.0 km/s
Plane Angle	-15.8^o
Solar Phase Angle	60.0^o
Earth Distance	0.07 AU

*EME2000

Table 3: DART Low Energy Escape Sample Mission

Component	Mass (kg)
Maximum Expected Dry Mass	463.6
Dry Mass Margin	63.3
Margined Hydrazine	31.2
Subtotal: Spacecraft Neutral Mass	558.1
Max Det. Xenon Over Launch Study	66.1
Operational Xenon Margin (3%)	2.0
Missed Thrust Xenon Margin (10%)	6.6
Impact Timing Adjustment	0.5
Startup Xenon Losses	0.3
Xenon Tank Residuals	3.0
Subtotal: Xenon Loading	78.5
Launch Mass	636.6

Table 4: DART Mass Budget for Low Energy Escape

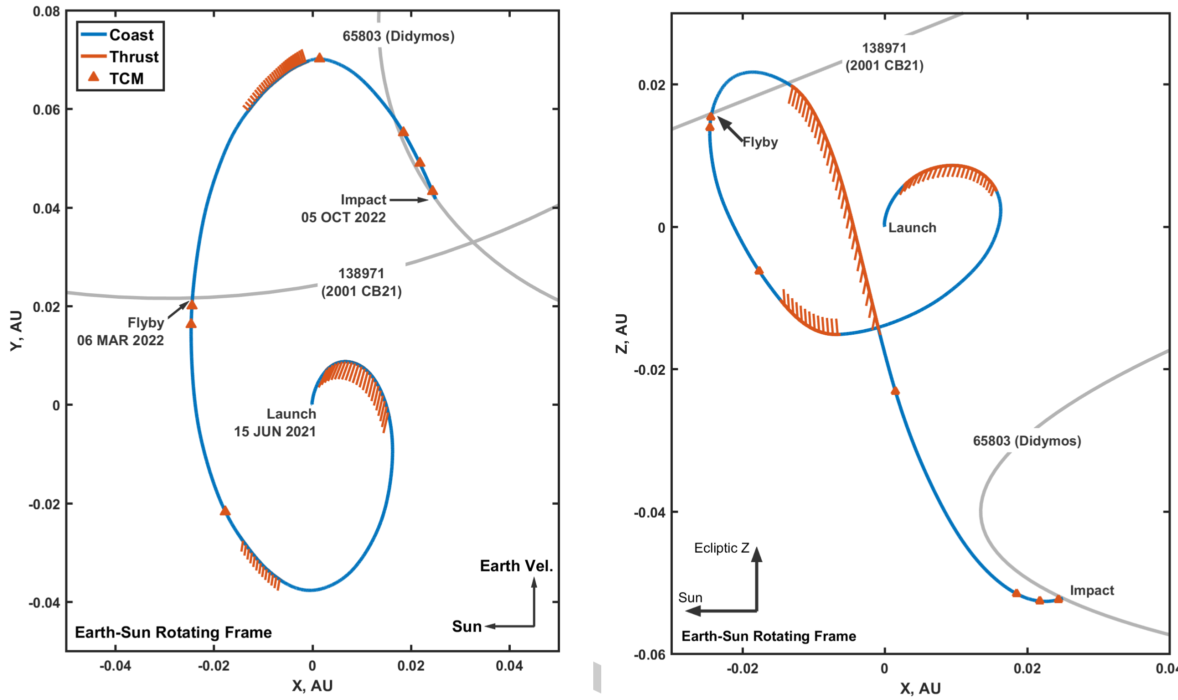

Fig. 5: Sample low-energy escape trajectory, given in Earth-Sun rotating coordinates.

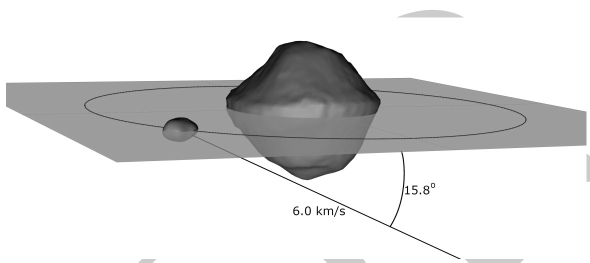

Fig. 6: Sample DART Impact Geometry.

GTO co-manifest partner. DART needs to be compatible with both cases.

Over a unit sphere, some of the escape directions will be favorable with respect to the underlying dynamics, while others will be in the opposite direction. To minimize the impact of the unfavorable directions, we have selected a low launch escape energy: C3 = +0.1 km^2/s^2. This energy is just high enough to provide a trajectory that deterministically departs the Earth-Moon system, without being so high that DART can't recover from an unfavorable direction. This is especially relevant since DART isn't anticipated to use its IPS until 7 days after launch. During this period, it will potentially be drifting away from Earth in an unfavorable direction.

One key study assessed the definition of C3. That is, C3 is a measure of orbit energy in a simplified two-body model. Its value osculates due to any additional perturbations, namely the Sun and Moon. Figure 7 gives a time-history of C3 from an initial altitude of 10,000 km to the Earth's sphere-of-influence (~950,000 km). Each line on the plot begins at the same time, and has a final C3 of 0.1 km^2/s^2. The initial values, which would be provided by the launch vehicle, vary from values lower than -0.1 km^2/s^2 to as high as 0.28 km^2/s^2. The evolution is primarily associated with the influence of the Moon and Sun. In some cases, lunar encounters significantly perturb the outbound energy. This study highlights the importance of defining and evaluating the launch energy at the right place, the Earth's sphere-of-influence. This ensures that DART and any associated debris depart the system.

With a consistent definition of C3, we studied the range of DART trajectory variation over different launch directions. Figure 8 shows a contour plot of deterministic xenon requirements for a range of escape asymptote directions. Higher values (in yellow) represent unfavorable escape directions that require longer amounts of thrusting to complete the mission. In each case, the C3 at the Earth sphere-of-influence was held constant at 0.1 km^2/s^2. There is a higher density of points between $\pm 30^o$ to better characterize

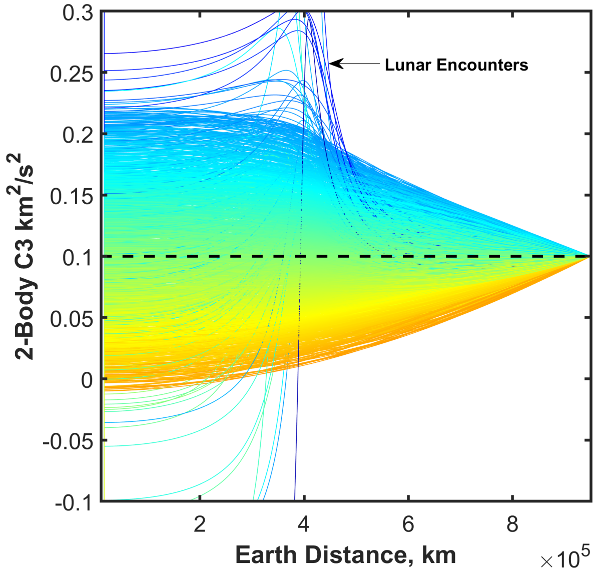

Fig. 7: Family of Two-body C3 Time Histories with values of 0.1 km^2/s^2 at Earth's Sphere-of-Influence.

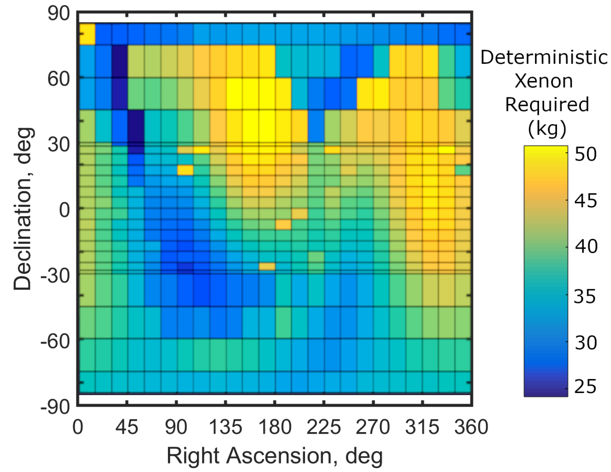

Fig. 8: Deterministic Xenon Requirements for Full DART Trajectory, as a Function of Initial Launch Escape Asymptote (EME2000) on the Launch Period Open.

potential US launched GTO co-manifest cases. Based on this analysis, the worst-case departure direction requires 50.7 kg of xenon to achieve the DART trajectory requirements. This is the case for which the baseline trajectory (Figures 4-5 and Table 3) was designed. This is the most conservative approach for the launch window open date.

Launch Period

The above analysis is focused on the first day in the mission's launch period. An important consideration is the determination of the latest possible launch date. This defines the available mission schedule and restricts the available launch co-manifest partners. Additionally, the DART spacecraft must be designed to accommodate variation in the trajectory throughout the launch period.

Of course, the latest launch date is related to the escape asymptote direction. For the most favorable directions, it's possible to launch as late as December 2021 and still achieve the baseline flyby of 2001 CB_{21}. Without the flyby, it's possible to launch as late as February 2022. However, since the mission cannot guarantee a favorable direction, we conducted an analysis to quantify the available launch directions as a function of launch date. Here, the xenon loading is held constant, consistent with Table 4. The only variability is the ability to achieve the flyby prior to impact. Figure 9 shows the baseline latest launch date, Oct 15 2021. For this date, $\sim 2/3$ of the studied directions are able to complete the mission with the flyby. If the mission were provided a launch on this date, we would seek to collaborate with the co-manifest partner and the launch vehicle to try to shift the launch direction to a favorable value. If this weren't possible, the mission would forgo the nominal flyby and transfer directly to impact.

Figure 10 depicts some of the variability across the launch period. The figure gives solar range as a function of days until impact for a variety of cases. Each case has a particular launch day (between Jun 15 2021 and Oct 15 2021) and direction (over the unit sphere). The color of the line correlates to the launch date. Portions of the lines that are thick indicate points when thrusting is occurring. For most cases, there are three thrust arcs, though some have only two. The lines converge at the impact, which occurs on the same date, and at the flyby, which has a variable date. This type of analysis is relevant for ensuring thermal and power performance across the full range of possible launch conditions.

Margins

Given the desired flexibility and practical uncertainty of the mission at this phase, we are continuing to account for generous margins where possible.[3,9] This includes the following:

- **Duty Cycle -** We maintain 5% unallocated duty cycle margin.

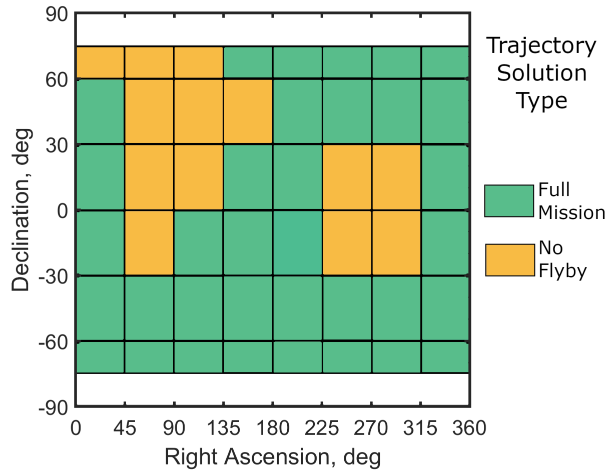

Fig. 9: Mission Profiles Possible for an Oct 15 2021 Launch, as a Function of Escape Asymptote Direction (EME2000).

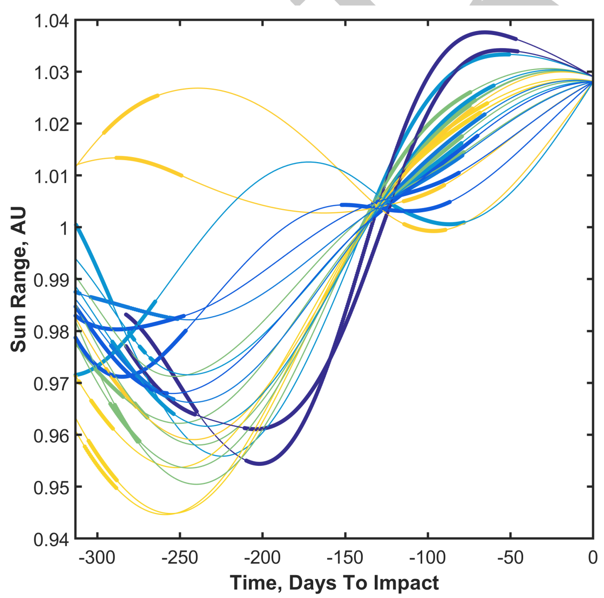

Fig. 10: Sample of Trajectories over Launch Period.

- **Specific Impulse** - Although DART will use only a fraction of the tested NEXT-C throughput, we use the end-of-life specific impulse values, which gives an effective 1.4% reduction in performance.

- **Operational Xenon** - We carry 3% unallocated xenon margin, which is intended to account for execution errors (attitude knowledge/control, thruster/gimbal pointing error, and thruster under-performance) and prediction errors (attitude prediction error and ephemeris prediction error).

- **Missed Thrust** - We include 10% xenon margin for recovering from a missed thrust event. The most sensitive portion of the flight is the first thrust arc, which must recover from any direction of escape asymptote. Figures 11 and 12 show an attempt to quantify one such case. Here, the launch period open is considered. Each point seeks to maximize the available duration between planned IPS operation (7 days post-launch) and required initial thruster operation, given a fixed xenon loading (per Table 4). Figure 11 shows a plot for 2 kg of available xenon for recovery. The cumulative statistics for this, as well as the 0 kg and 5 kg cases are given in Figure 12. As an example point, 90% of the launch cases are recoverable in 17, 19, and 23 days for 0, 2, and 5 kg of available xenon propellant respectively.

Additionally, all of the hydrazine is carried to the end of the mission, without accounting for use throughout the mission for attitude control and statistical maneuvers. Likewise, the mission closes with all of the explicit margins and statistical xenon entries unused. These two assumptions mean that the IPS must carry a higher effective neutral mass to the final encounter.

IV. Low Energy Escape - Navigation

The DART navigation architecture flies out the trajectory provided by mission design, using orbit determination techniques and targeting maneuvers to arrive at Didymos. Figure 13 presents the relevant mission phases. Radiometric measurements, in the form of 2-way Range and Doppler, and optical navigation (OpNav) measurements are used to estimate DART's trajectory in flight. The radiometric measurements are provided by the DSN while OpNav measurements are taken on approach to 2001 CB_{21} and Didymos using DART's DRACO camera.

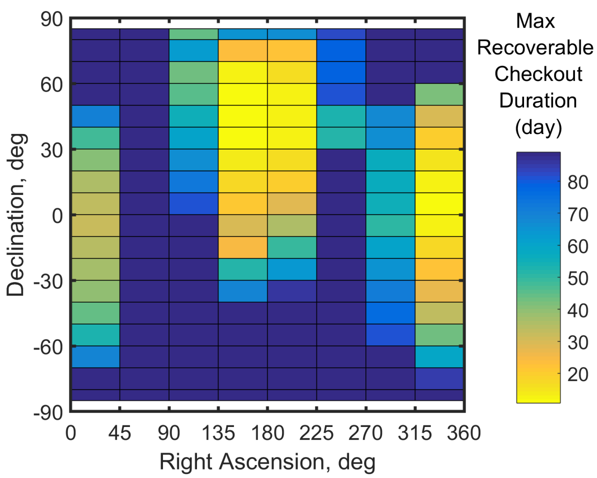

Fig. 11: Maximum Recoverable Checkout Duration for 2 kg of Propellant Allocation, as a Function of Right Ascension and Declination (EME2000) over the Launch Period Open Date.

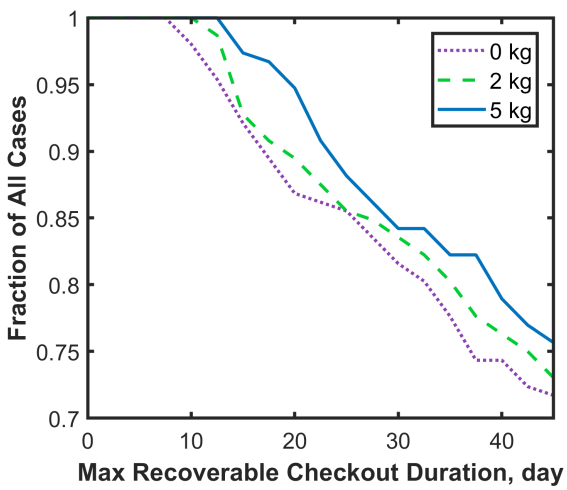

Fig. 12: Cumulative Statistics for Maximum Recoverable Checkout Duration on the Launch Period Open.

The cruise navigation support begins following launch vehicle separation and ends 12 hours prior to Didymos-B impact, when the onboard SmartNav system is activated for terminal targeting. Navigation's primary objective is to deliver DART to the Didymos system within the targeting tolerances required by SmartNav. The requirement is to deliver DART on a terminal trajectory targeted to fly within 15 km ($1\ \sigma$) of the Didymos system center. Once activated, the SmartNav system uses the DRACO imager to perform onboard optical navigation during the terminal phase, steering DART to its final target.

Navigation features the use of both low thrust burns and statistical TCMs to target Didymos, using the NEXT-C ion propulsion system and the attitude control system thrusters, respectively. The baseline trajectory features three low thrust electric propulsion (EP) arcs. Broadly speaking, EP Arc 1 is used to achieve proper orbit phasing following launch, EP Arc 2 targets the 2001 CB_{21} asteroid flyby, while EP Arc 3 targets the Didymos system. Since the low thrust burns provide neither the accuracy nor the control authority required to meet approach targeting objectives, they are augmented with chemical TCMs. TCM1 and TCM4 and placed 10 days following engine cutoff for EP Arc 2 and EP Arc 3, respectively. These maneuvers are primarily used to clean up execution errors from the EP Arcs soon after engine cutoff. TCM2 and TCM5 are placed a few days after each OpNav campaign begins in order to correct targeting errors due to the asteroid ephemeris prediction error. The remaining TCMs provide correction opportunities in the final days of approach as the OpNav measurements of each asteroid become stronger. The current baseline includes six TCMs, with one contingency backup maneuver.

At 12 hours prior to the flyby and impact events, the cruise navigation team uplinks an ephemeris prediction to DART and activates SmartNav for onboard terminal navigation. The 2001 CB_{21} flyby is intended to be a rehearsal opportunity for the impact, and thus will exercise SmartNav measurement processing with thrusting disabled.

IV.i Maneuver Design Concept-of-Operations

The DART maneuver architecture is intended to balance operational complexity with propellant needs. Ideally, each EP arc could be fully executed without any interaction with the ground. However, the buildup of burn execution errors during each EP arc would drive future TCM cleanup maneuvers to be unacceptably large. This is especially true consid-

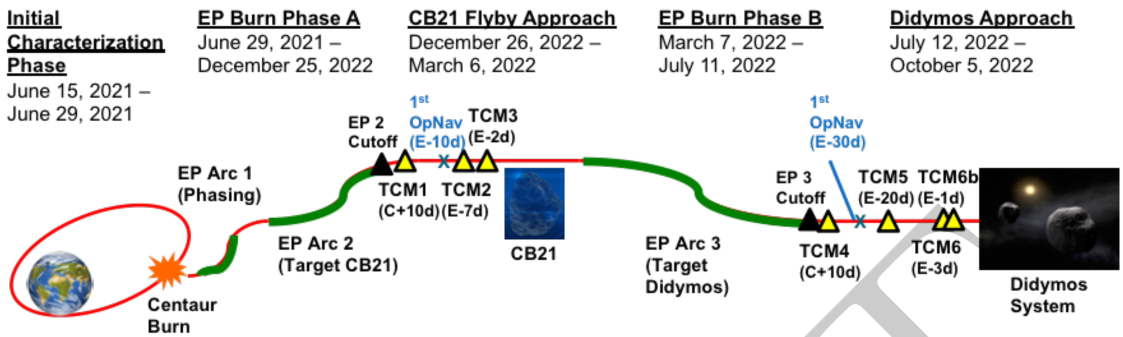

Fig. 13: Overview of DART Mission Phases.

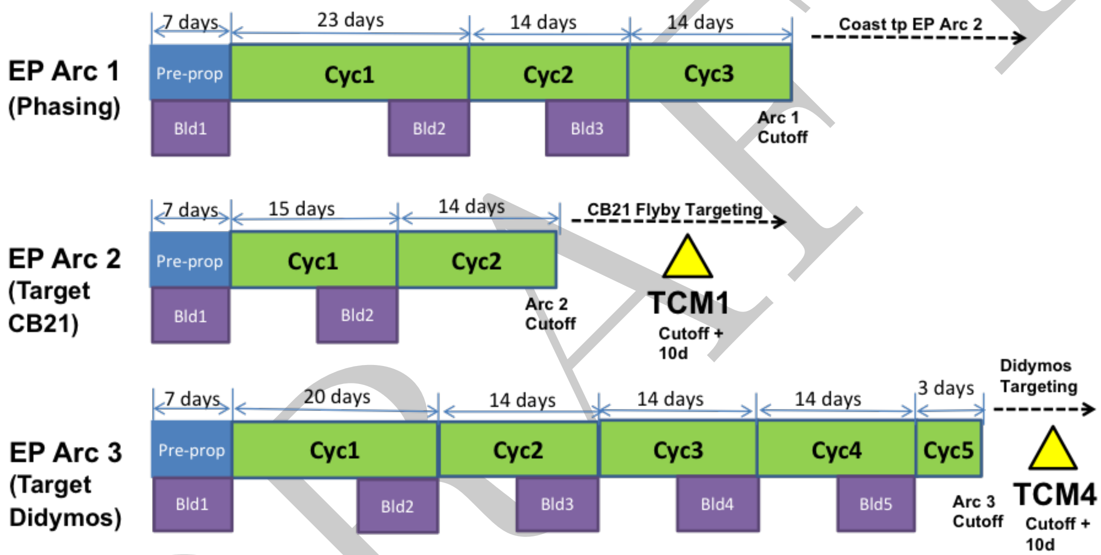

Fig. 14: Low Thrust Burn Architecture.

ering each EP arc in the baseline trajectory features continuous thrust for 1-2 months in duration with execution errors accumulating throughout the burn.

To solve this problem, each EP arc is segmented into design cycles that are generally 2-3 weeks in duration, as depicted in Figure 14. The build blocks represent 7-day build periods when the next burn cycle is designed on the ground. The build block begins with the orbit determination (OD) data cutoff (DCO) and concludes when the designed cycle begins execution on the spacecraft. The cycle blocks represent the periods of active thrusting on the spacecraft.

This architecture mitigates the buildup of large trajectory errors since each cycle design can correct errors accumulated in the previous thrust cycle (up to the build start). For example, errors accumulated during the first 7 days of cycle 2 thrusting will be corrected in the cycle 3 build and flyout. Therefore, the majority of trajectory error at each EP arc cutoff is driven by only the final thrusting cycle. The assumed burn execution errors are 0.5% on the thrust magnitude and 0.25^o (1σ) stochastic pointing bias per thrusting arc. This is the level assumed following thrust calibration, which occurs during the first two weeks after launch.

The sizing of cycle durations was determined by examining the impact on downstream TCMs, which have a limited hydrazine budget. The Delta-V (DV) allocation to navigation for statistical TCMs is 20 m/s for all maneuvers. Initially, each cycle was sized at one month in duration each, resulting in a total TCM DV of 51.4 m/s. Reducing the cycle duration to 14 days mitigated the propellant issue while still keeping operations relaxed to an every-other-week de-

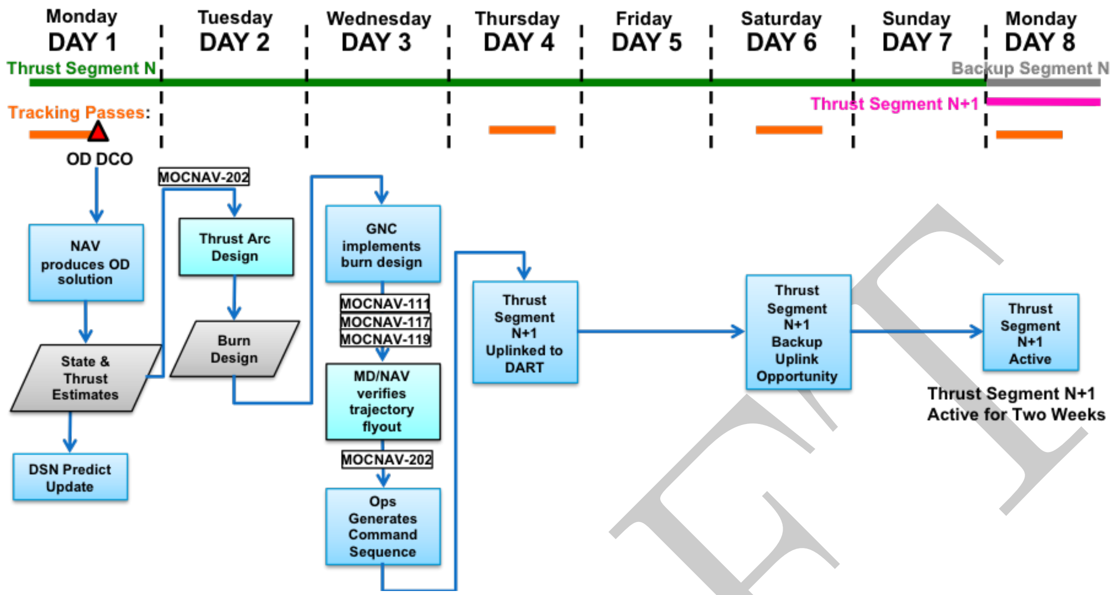

Fig. 15: Low Thrust Maneuver Build CONOPS.

sign cycle. However, TCM4 remained unacceptably large in this architecture. The solution here was to create a three-day "mini" thrust cycle at the end of EP Arc 3 to correct accrued errors. The final 99th percentile DV for this solution varies from 11 m/s and 16 m/s depending on error scenario assumptions, as will be discussed in greater detail in Section IV.ii.

Figure 15 describes the concept of operations for each low thrust maneuver design build period. The build cycle begins once the final radiometric tracking pass prior to DCO terminates, nominally on the first day of the build week. From the latest measurements and updated spacecraft dynamical model, the navigation team uses a linearized least-squares filter to estimate the spacecraft states and thrust profiles with associated errors, to then provide an updated trajectory prediction for both the DSN and the mission design team. On day 2, the initial thrust cycle design is produced by the mission design team. The guidance and control team delivers an attitude pointing sequence on day 3 that implements the designed thrusting profile. This sequence is ingested into navigation tools and propagated to validate the thrust cycle implementation. If validation passes, the thrust profile is generated into a command sequence and uplinked to DART on day 4, with a backup opportunity on day 6. On day 8, the next cycle commences thrusting.

The TCM build cycle, in contrast, is more stressing due to the quick turnaround times required between OpNavs and maneuver execution. The TCM build durations are as long as 3 days for TCM1 and TCM4 and as short as 12 hours for TCM3 and TCM6 since these maneuvers occur within only a few days of the flyby and impact events. Note that TCM6b is a contingency maneuver to be implemented only if TCM6 doesn't execute. Figure 16 depicts the TCM build concept of operations for the shortest build cycles. The last OpNav measurement incorporated into the TCM design occurs three hours prior to the radiometric tracking DCO, since camera images must be downlink from the spacecraft and processed. Navigation is allocated three hours for producing an orbit determination solution using both radiometric and optical data, which is then passed on to the maneuver team to generate a burn solution. From those maneuver products, the attitude control subsystem implements a slew and thrust design. Similar to the low thrust concept of operations, the trajectory flyout is validated prior to maneuver uplink to the vehicle.

IV.ii Navigation Targeting Results

The architecture described in the previous two sections is assessed for compliance against navigation targeting requirements in this section. Navigation performance is modeled using JPL's Mission Operations and Navigation Toolkit Environment (MONTE), which provides flight dynamics models, navigation error models, and the capability to simulate and process measurement data. Tables 5 & 6

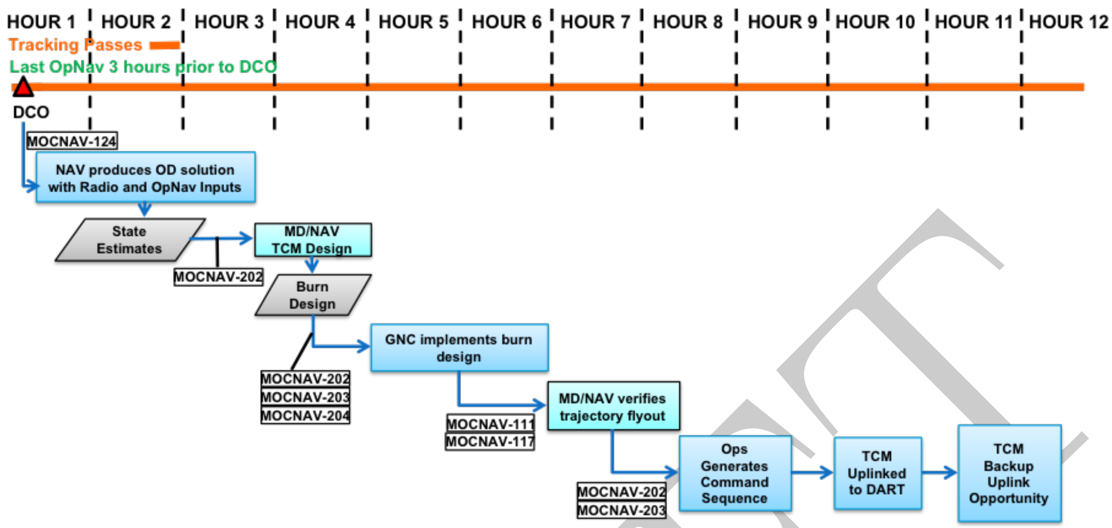

Fig. 16: Trajectory Correction Maneuver Build CONOPS (Shortest Cycle).

Parameter	Uncertainty
Spacecraft Position	100 km
Spacecraft Velocity	100 m/s
Solar Pressure	Not-Modeled
EP Force Bias	0.6855 mN (0.5%)
EP Pointing Biases	0.25^o RA, 0.25^o Dec
TCM Delta-V	5 cm/s spherical
Pole Motion	15 nrad
UT1	0.25 msec
Media Calibrations	0.01 m Tropo, 0.55 m Iono

Table 5: Navigation Error Models: Dynamic & Bias Parameters

Parameter	Uncertainty
Non-Gravitational Acceleration	1e-10 km/s^2 1 hr batches
Range Measurement Bias	2 m

Table 6: Navigation Error Models: Stochastic Parameters

overview the various error models used in the OD filter to assess the DART navigation architecture performance. The dominant error contributors include the EP execution errors, the TCM execution errors, and the non-gravitational acceleration disturbances induced by the attitude control system. The execution errors are modeled as per-cycle bias errors that can be estimated over time, while the ACS disturbances are modeled as fully stochastic white noise.

Navigation performance is assessed by inspecting the B-plane targeting error as a function of maneuver location. The B-plane is a geometric representation of the nominal intersection of DART's trajectory with the Didymos system. The B-plane delivery error is calculated by mapping navigation errors from each maneuver DCO to the nominal impact time. Figure 17 depicts the B-plane delivery error with the maneuver DCOs highlighted with dark vertical lines.

Without any TCMs, the last cycle of EP Arc 3 would deliver DART to the Didymos system with 2590 km (1σ) cross track targeting error and 1310 seconds (1σ) of time-of-flight error. TCM4, the first statistical TCM following EP Arc 3, reduces the delivery error to 584 km (1σ). Following TCM4, the improvement in delivery error is largely a result of DSN tracking resolving the execution errors from EP Arc 3 and TCM4. The targeting performance levels off to 130 km (1σ) as the asteroid ephemeris uncertainty becomes the dominant error source in the prediction. Following the first OpNav measurements, TCM5 is able to reduce delivery error further to 102 km (1σ). The OpNav measurements provide stronger insight into the asteroid ephemeris as the range to Didymos closes, resulting in more accurate deliveries by TCM6 (13 km). We include a contingent backup maneuver, TCM6b one day before impact if TCM6 cannot be executed. TCM6 meets the required delivery requirement with approximately 15% margin.

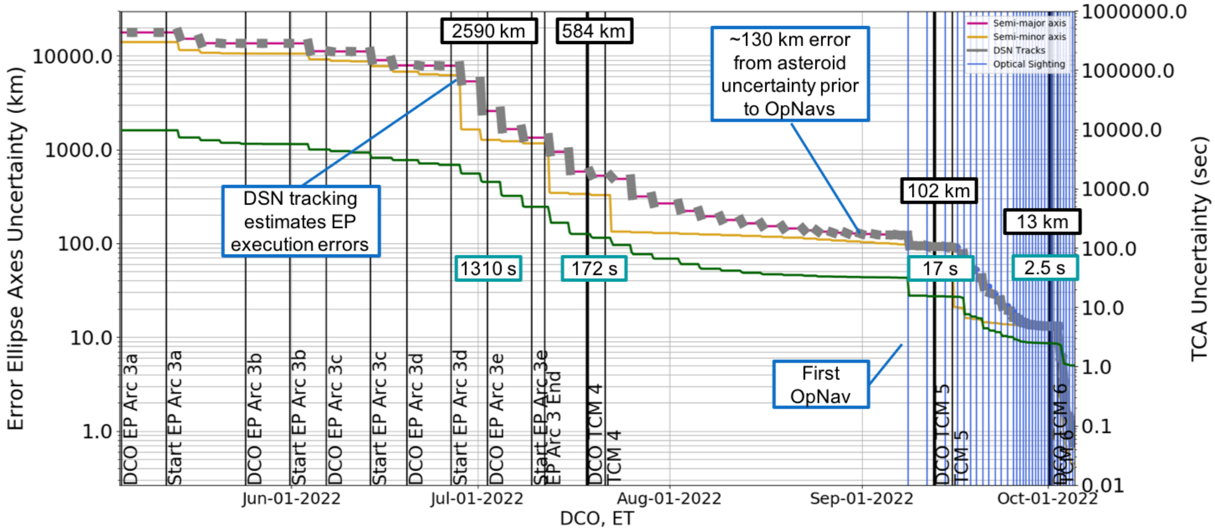

Fig. 17: Didymos B-Plane Targeting Error as a Function of Maneuver Time. The thick gray bars superimposed over the semi-major axis plot represent DSN track periods, nominal two times per week and increasing in frequency as the impact date nears. The blue vertical lines represent OpNav measurement times.

IV.iii TCM Maneuver Results

In order to execute the delivery strategy described in the previous section, navigation must remain within the propellant budget allocated for TCMs. The propellant budget is sized using a Monte Carlo maneuver assessment. The covariance at each maneuver DCO is sampled, mapped forward to the Didymos B-plane, and a Delta-V maneuver to target the B-plane center is computed. This process is repeated for 10,000 samples and the DV statistics for each TCM opportunity is computed. Table 7 shows the statistical DV results for each TCM. TCM6b is not included, since this maneuver is contingency only and will not be executed unless TCM6 does not fire. The 99th percentile result for all TCMs is 9.46 m/s for targeting 2001 CB_{21} and Didymos, assuming the nominal EP errors stated in the previous section. The majority of propellant is spent on TCM1 and TCM4, reflecting the influence of EP Arc execution errors on the TCM budget. The remainder of TCMs primarily correct for asteroid ephemeris errors as well as previous TCM execution errors.

Maneuver	Time	Mean (m/s)	Sigma (m/s)	99% (m/s)
TCM1	C + 10d	2.47	1.08	5.39
TCM2	E - 7d	0.84	0.39	1.95
TCM3	E - 2d	0.30	0.13	0.62
TCM4	C + 10d	1.00	0.63	2.93
TCM5	E - 20d	0.63	0.35	1.69
TCM6	E - 3d	0.56	0.24	1.20
Total		5.80		9.46

Table 7: Maneuver Targeting Delta-V Results

V. Conclusions

DART is the first mission to attempt an impact with a 100-m class asteroid for the purposes of planetary defense. The mission has considered a variety of different implementations and launch vehicles. The low energy escape concept, enabled by the addition of the NEXT-C thruster, offers a low cost design that is compatible with a co-manifest partner.

The flexibility of the IPS allows for a wide range of launch conditions and dates. The baseline launch period extends from Jun 15 2021 to Oct 15 2021, with opportunities to launch many months later if a favorable launch direction is provided or if the opportunistic flyby of 2001 CB_{21} is removed.

To enable this mission, DART navigation delivers the spacecraft to the Didymos system with sufficient accuracy for the SmartNav onboard terminal guidance to impact Didymos-B The navigation architecture features a series of low thrust maneuver cycles using the Next-C ion propulsion engine. The use of low thrust provides both an increased Delta-V efficiency and increased flexibility in the navigation execution. Each cycle design of a low thrust maneuver can correct errors accrued in the previous cycle, resulting in constant refinement of the spacecraft trajectory. During the final 45 days of approach to 2001 CB_{21} and Didymos, chemical TCMs are executed for final refinement of the trajectory prior to SmartNav handoff.

VI. Acknowledgments

This work was funded under the NASA Double Asteroid Redirection Test program (Contract NNN06AA01C), through the Planetary Defense Coordination office. Part of this research was carried out at the Jet Propulsion Laboratory, California Institute of Technology, under a contract with the National Aeronautics and Space Administration.

References

[1] Andy F. Cheng, J. A. Atchison, Brian Kantsiper, Andrew S. Rivkin, Angela M. Stickle, Cheryl Reed, Andres Galvez, Ian Carnelli, and Patrick Michel. Asteroid impact and deflection assessment mission. *Acta Astronautica*, 115:262–269, 2015.

[2] Brian Kantsiper, Andy Cheng, and Cheryl Reed. The double asteroid redirection test mission. In *Aerospace Conference, 2016 IEEE*, pages 1–7. IEEE, 2016.

[3] Bruno V. Sarli, Martin T. Ozimek, J. A. Atchison, Jacob A. Englander, and Brent W. Barbee. NASA double asteroid redirection test (DART) trajectory validation and robustness. *AAS Spaceflight Mechanics Conference, San Antonio TX, AAS-17-206*, 2017.

[4] Zachary J Fletcher, Kyle J Ryan, BJ Maas, JR Dickman, RP Hammond, DL Bekker, TW Nelson, JM Mize, JM Greenberg, WM Hunt, et al. Design of the didymos reconnaissance and asteroid camera for opnav (DRACO) on the double asteroid redirection test (DART). In *Space Telescopes and Instrumentation 2018: Optical, Infrared, and Millimeter Wave*, volume 10698, page 106981X. International Society for Optics and Photonics, 2018.

[5] Michelle H. Chen, Justin A. Atchison, David J. Carrelli, Peter S. Ericksen, Zachary J. Fletcher, Musad A. Haque, Stephen N. Jenkins, Mark A. Jensenius, Nishant L. Mehta, Timothy C. Miller, Daniel O'Shaughnessy, Carolyn A. Sawyer, Emil A. Superfin, and Richard D. Tschiegg. Small-body autonomous real-time navigation (SMART Nav): Guiding a spacecraft to didymos for nasa's double asteroid redirection test (DART). In *AIAA Guidance, Navigation, and Control Conference, AAS 18-081*, pages 1–13, 2018.

[6] Brian Kantsiper. The double asteroid redirection test (DART) mission electric propulsion trade. In *IEEE Aerospace Conference, Big Sky MT*, pages 1–7. IEEE, 2017.

[7] J. A. Atchison, Martin T. Ozimek, Brian L. Kantsiper, and Andrew F. Cheng. Trajectory options for the DART mission. *Acta Astronautica*, 123:330 – 339, 2016.

[8] Martin T. Ozimek and J. A. Atchison. NASA double asteroid redirection test (DART) low-thrust trajectory concept. *AAS Spaceflight Mechanics Conference, San Antonio TX, AAS-17-221*, 2017.

[9] David Y. Oh, Damon Landau, Thomas Randolph, Paul Timmerman, James Chase, John Sims, and Theresa Kowalkowski. Analysis of system margins on deep space missions utilizing solar electric propulsion. In *AAS 2008-5286, Hartford CT*, pages 1–30, 2008.